Bonded Repair Technology for Aging Aircraft

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Abstract

NATO weapons systems are being extended well beyond their design lives due to decreases in defence budgets and the rising costs associated with procuring new hardware. This situation makes it increasingly important that methods for extending the lives of these weapon systems in a cost-effective manner be developed and implemented to the greatest extent possible. Adhesive bonding technology, particularly bonded composite repairs/enhancements, has been successfully applied by several nations to extend the lives of aircraft by bridging cracks in metal structure, reducing strain levels, and repairing areas thinned by corrosion. Bonded composite reinforcements are highly efficient and cost effective when compared to conventional mechanically fastened approaches. In some cases, bonded repair technology is the only alternative to retiring a component. This technology has already resulted in the documented savings of hundreds of millions of dollars in Australia and the United States.

This paper describes the advantages of bonded composite repairs over conventional repair methods. Bonded joint design/analysis, installation procedures, nondestructive inspection, certification issues, and other key aspects of the technology are generally addressed. Examples of applications to aircraft are used to illustrate these issues as well as demonstrate bonded repair advantages. The capabilities and resources required to successfully apply bonded repairs are discussed. Finally, several recent reviews of this technology area are summarised to indicate where the key scientific gaps remain and to suggest research that should be undertaken to further enhance the usefulness of the technology.

1. INTRODUCTION

Military aircraft are increasingly being required to remain in service for times that are longer than their original design lives. This is due to both decreasing defence budgets and high costs for replacement aircraft. As the aircraft age, managers will face ever-increasing amounts of damage that will require repair. Conventional repair methods typically involve either replacement of the damaged component or installation of a

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mechanically fastened repair. Both approaches are well established but suffer from lengthy aircraft down times and high costs. In addition, mechanically fastened repairs may not be viable for certain applications. The approach of using adhesive bonding to repair or reinforce damaged aircraft structure has been shown to be a highly cost effective alternative to the conventional repair methods [1]. Thousands of adhesively bonded repairs have now been applied to hundreds of aircraft in service with the Royal Australian Air Force (RAAF) [2] and United States Air Force (USAF) [3] since the middle 1970s.

Mechanically fastened repairs are usually less expensive than component replacement, however, the need to drill new holes for the fasteners in the structure is a major limitation. These holes will act as stress concentrators and may result in the initiation of new fatigue cracks. As a mechanically fastened repair transfers load only through the fasteners, it is not particularly stiff and so the damage must usually be removed from the structure. Adhesively bonded repairs are much more efficient due to the uniform load transfer mechanism and repairs can therefore typically be made to untreated or "live" cracks. Validated, predictive methods now exist to calculate the crack growth rate after repair to assist with certification. By avoiding the need to remove the damage, adhesively bonded repairs are much less intrusive, are less likely to cause unexpected damage (to hidden wiring or hydraulics for example) and are faster to apply.

The technology can also be used very effectively to reinforce undamaged structure that is known to be under designed and in danger of developing fatigue cracks, for example, at some later stage. In this regard, the technology is highly effective in extending the life of an aging aircraft structure.

As the technology has matured and repairs have become more routine, the aircraft operators have required the development of rigorous engineering standards to underpin the application of repairs. Importantly, the technology has also been incorporated within aircraft structural integrity programs that provide the required level of engineering management. This includes issues such as training of staff in design and application procedures, quality control and configuration control. Successful use of the technology requires an overarching engineering management framework such as this.

Adhesively bonded repairs are routinely applied to tertiary and secondary structure, and there have been significant applications to safety-of-flight-critical (primary) structure. Research is now focusing at making these repairs even easier to design and apply. It is also addressing the certification concerns that currently prevent aircraft maintainers from taking full benefit of the technology on primary structure applications.

2. DESIGN

2.1 Design Approaches

The first requirement is to assess the defect, assumed here to be a crack, in terms of its length and depth, and to determine the thickness and geometry of the cracked region as well as the local loading conditions. Of particular importance in adhesively bonded repairs is the available overlap length on either side of the crack. The thicker the structure and consequently the higher the loads, the longer the overlap length needed to transfer the loads into the patch. Highly loaded repairs require appropriately thick patches

[4] a) to provide adequate reinforcement, b) to prevent failure of the patch and c) to prevent failure of the adhesive.

In the most usual case where the repair can be applied to only one side of the structure, the degree of support of the structure against secondary bending must also considered. Secondary bending which results from the displacement of the neutral axis of the parent structure by the patch can markedly change the loads experienced by the patch and structure [5].

The temperature and environment experienced by the region to be repaired must also be considered, since this will help determine the type of adhesive to be used. Adhesive choices are considered later.

It is unlikely, other than for relatively simple cases such as fuselage repairs, that information on local loading will be available, unless there is access to detailed design data. Failing this, a good approximate estimate of the stress at design limit load (DLL) is achieved by equating design ultimate load (DUL) with the material yield stress σ_y. The basis for this is that a far-field stress level equal to material yield stress failure could occur for aluminium alloys at regions where Kc > 1.2. Then DLL is the yield stress divided by 1.5. This has proven to be a conservative estimate in all cases examined by one of the authors (AAB) where the DLL was known. Table 1 is an example of a correlation. Clearly, assumption of the ultimate strength σ_u as equal to σ_{DLL} is over conservative.

DADTA Item No	σ_{DLL}	σ_Y	σ_u	σ_u/σ_Y	$\sigma_Y/1.5$	σ_Y/σ_{DLL}	$\sigma u/\sigma_{DLL}$
67	202.9	400.2	462.3	1.2	266.8	1.97	2.28
70	167.0	400.2	462.3	1.2	266.8	2.40	2.77
70a	204.2	400.2	462.3	1.2	266.8	1.96	2.26
78	149.7	400.2	462.3	1.2	266.8	2.67	3.09
154	171.8	400.2	462.3	1.2	266.8	2.33	2.69
194	165.6	400.2	462.3	1.2	266.8	2.42	2.79

Table 1: *Data on Design Limit Stress σ_{DLL} for F-111 for several (DADTA) data points in the lower wing made of aluminium alloy 2024-T581, compared with the yield stress σ_y.*

A knowledge of the loading spectrum for the region is also unlikely to be available, so the best approach, if such detail is required for design, is to assume one of the standard spectrums, FALLSTAF or TWISS, according to whether the aircraft is a fighter or transport.

Once sufficient information is available concerning the loading and other parameters and it is considered on the basis of the forgoing discussion that a bonded repair is feasible, a patch design can be undertaken. For patch design, there are two basic approaches: analytical usually based on the Rose model [6] for simple loading and geometries or, for more complex geometries or loading, finite element analysis. CalcuRep, developed by the USAF Academy, is an established software program for the analytical approach. Earlier, an iterative approach was developed by Baker [4] to determine optimum patch dimensions. In patch design, the aim is to determine the reduction of stress intensity experienced by the repaired crack as well as the stress levels in the patch and adhesive system. The goal is to avoid failure of the patch system, while providing a sufficient reduction in stress intensity to ensure the required service lifetime for the repair.

Fatigue studies [7] were undertaken to validate use of the Rose model for patch design. Generally, it was shown that satisfactory prediction of patching performance could be made over a range of variables, including stress patch stiffness and R ratio based on the usual semi-empirical crack growth equations and the predicted the stress intensity. It was also found possible to extend the model to include allowance for disbond growth. Figure 1 shows a standard log/log plot for crack growth versus predicted ΔK for a range of patch thicknesses.

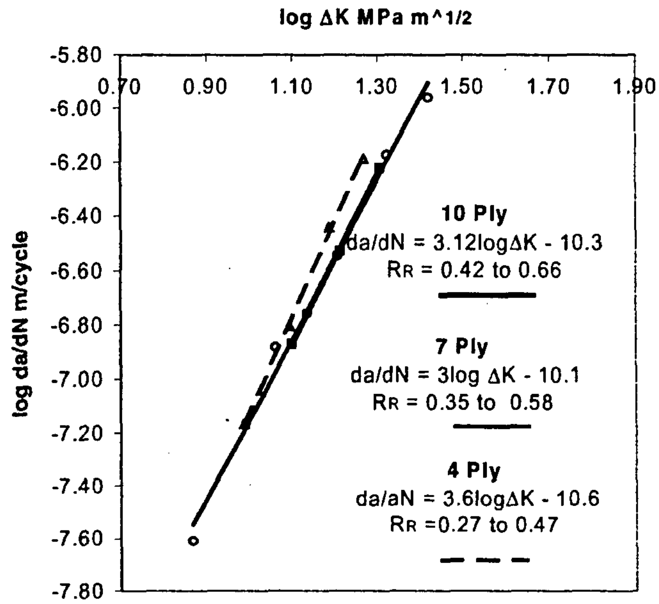

Figure 1: *Plots of crack growth rate da/dN versus log ΔK for various thickness boron/epoxy patches.*

2.2 Patch and Adhesive Choices

For the patch material, there are three main options generally considered: the fibre composites boron/epoxy and carbon/epoxy and the aluminium alloy-glass/epoxy laminate GLARE. The need is for high strength and stiffness, fatigue and environmental durability and formability. The composites satisfy most of the requirements; however, their main disadvantage is their low thermal expansion coefficient which gives rise to undesirable residual tensile stresses in the repaired component.

Most Australian and U.S. repairs to date have used boron/epoxy as the reinforcement rather than graphite/epoxy by virtue of its superior properties for this application:

- Better combination of strength and stiffness.
- Electrically nonconductive, avoiding galvanic corrosion problem with aluminium and facilitating eddy-current nondestructive inspection (NDI) of cracks under the repair.
- Higher coefficient of thermal expansion (CTE), minimising residual-stress problems.
- Better fibre alignment under cocure conditions as a result of much larger fibre diameter — 125μm compared with 8μm for graphite fibres.

However, compared to carbon/epoxy, boron/epoxy is much more costly, less readily available and because of the large fibre diameter less formable. Thus carbon/epoxy is used whenever it is more cost effective or where very high formability is required. Work by Poole [8] in the UK, has shown that carbon/epoxy provides very satisfactory properties as a patch material.

GLARE (aluminium/fibreglass laminate) is an alternative patch material that has high fatigue resistance and important benefits [9] where minimising residual stresses is important; however, it has limited formability and relatively low stiffness so is best suited to the repair of thin-skinned fuselage components.

The optimum choice for the adhesive is generally a structural epoxy film. The main adhesive used in Australian repairs is Cytec Fiberite FM 73, a nominally 120°C-curing epoxy-nitrile structural film. This adhesive, or a similar epoxy film, is also most often used in the U.S.. Reasons for this choice include the following:

- Excellent strength and toughness from low to moderate temperatures
- Resistance to aircraft fluids.
- Ability to form strong durable bonds using appropriate prebond treatments.
- Ability to cure (with some sacrifice in properties) at relatively low temperatures - as low as 80°C (with extended times) compared with the standard one hour at 120°C.

The first three advantages are typical of most moderate-temperature-curing structural epoxy-nitrile film adhesives. However, the ability of FM 73 to cure at temperatures as low as 80°C is both unusual and valuable for repairs where the higher temperatures cannot be achieved or where there is a need to minimise residual stresses. For higher-temperature applications (above 80° C) the adhesive FM 300-2, also by Cytec Fiberite, is most often selected. This adhesive also has a capacity to cure at a relatively low temperature (120°C) while providing properties typical of a 175°C-curing adhesive. Finally, modified acrylic adhesives have been found to be highly effective for less demanding applications (temperatures not exceeding 60°C or not below –10°C, if peel stresses are high) or where the use of elevated cure temperature is not feasible. Some two-part epoxy paste adhesives may also be used when elevated-temperature curing is impractical or undesirable, but these are typically confined to non-critical applications.

2.3 Processing Choices

The processes by which the adhesive and patch materials are installed on the aircraft have a direct influence on the final properties and long-term durability of the repair. The material properties considered for design should take into account the effects of these processes, such as the cure cycle (time/temperature) and pressure application method used for adhesives and cocured patches. Selection of repair area and patch surface preparation processes is also an important design consideration. This task can be somewhat difficult since no tests exist that can quantitatively correlate laboratory test performance with service life. Prior experience and screening using the wedge test [10] have been used to select metal surface preparations for most Australian and U.S. repairs. However, it is important to note that the test specimen but not necessarily the moisture conditioning and acceptance criteria are per the referenced wedge test standard.

3. INSTALLATION

Successful bonded repair installation is not necessarily difficult but requires proper execution of a number of steps. These include surface preparation both for the aircraft structure and patch material as well as heating and pressurisation. Other considerations include the nature of the repair installation environment, handling of repair materials, health and safety issues, training of repair installers and post-bond operations.

3.1 Surface Preparation

Preparation of adherend surfaces prior to bonding is the single most important application process step for ensuring a successful repair [11]. Proper surface preparation is necessary to achieve initial bond strength and long-term durability in the service environment. Although the environment includes temperature extremes and exposure to many aircraft fluids and maintenance chemicals, moisture tends to be the biggest impediment to long-term durability, particularly for bonded aluminium joints [12].

A metal surface preparation typically must remove contaminants and naturally occurring oxide from the metal surface. It must also chemically and/or physically modify the surface to promote adhesion with the adhesive (or primer) and enable it to resist moisture attack. Cleaning and deoxidising alone can sometimes provide adequate initial adhesion but rarely result in bonded joints with long-term service lives. For repair applications, only very simple and nonhazardous treatments that can be applied on aircraft under field conditions are considered viable for most applications. Ideally, such a surface preparation will yield highly durable bonded joints with a variety of metal substrates.

To satisfy these requirements, Australian work is focused on the use of silane coupling agents. The coupling agent found most suitable for epoxy adhesives is the epoxy-terminated silane, γ-GPS [13]. This coupling agent provides high-strength durable bonds to aluminium alloys, stainless steel, nickel and titanium alloys. It is applied from an aqueous solution to the metal surface following mechanical conditioning by alumina grit blasting. The silane treatment is safe since it does not rely on noxious chemicals or electrical power. The process does not use acids, so it eliminates the corrosion concerns they cause if not properly rinsed and their potential to embrittle high-strength steel fasteners.

The durability against moisture degradation provided by the silane can be further enhanced by use of a standard corrosion-inhibiting primer. Figure 2 shows typical results for crack growth in the wedge test for 2024-T3 aluminium bonded with FM 73 adhesive following a) grit-blasting, b) grit-blasting + silane and c) grit-blasting + silane + primer. Although this test is considered by the authors to be the best available method to screen surface preparations, it is not ideal since test criteria that directly relate to bonded joint service cannot be established. The test can compare different surface preparations, with all other factors being equal, and relate the results to service experience. Minimal crack extension following exposure to hot/wet conditions and a crack that remains predominantly in the adhesive (rather than at a metal interface) indicate a good surface preparation. For FM 73 adhesive, conditioning at 50°C and 100% relative humidity (RH), crack growths of about 5 mm after 7 days of exposure tend to indicate an adequate surface treatment. This behaviour is exhibited by system c) in the figure. However, despite the longer crack extension and interfacial failure modes, system b) has performed adequately in service for many applications.

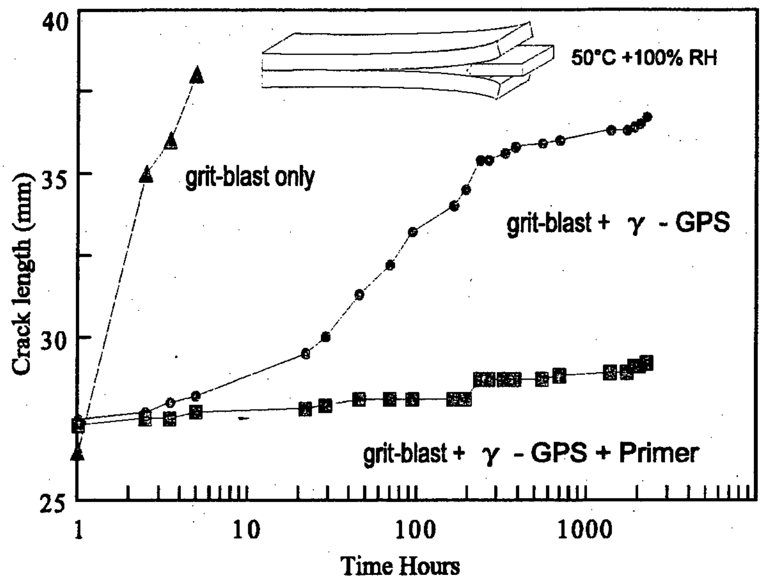

Figure 2: *Plots of crack growth versus time for wedge-test specimens (illustrated inset) made of 2024-T3 aluminium and subjected to the surface treatments indicated, prior to bonding with FM 73 adhesive.*

Most U.S. repairs have followed the Australian lead and employed a grit-blast/silane (GBS) surface preparation [14], including application of a corrosion-inhibiting adhesive bond primer, as a practical on-aircraft prebond treatment that yields good in-service environmental durability. Recently, a process based on sol-gel chemistry developed for the USAF by The Boeing Company has emerged for applications on the same metal alloys currently treated using GBS [15,16]. This approach is similar to GBS but employs a more reactive chemistry. The results are a simpler, quicker process that performs as well or better than GBS in laboratory tests, including the wedge test. A lesser-performing variant of the sol-gel approach that may be suitable for many noncritical applications eliminates the grit-blasting and priming steps [17].

For precured (thermosetting) fibre composite patches, surface removal by light grit blasting with alumina is a highly effective treatment that provides excellent bond strength and durability [18]. The standard peel-ply surface treatment procedure is not as effective unless followed by grit blasting or some other effective mechanical method of surface removal. Often, a cocured layer of adhesive is applied to the surface of the boron/epoxy patch to increase the toughness of the surface resin and to provide a layer more suitable for grit blasting. The surface of the as-received GLARE material is a cured corrosion-inhibiting adhesive bond primer. Solvent cleaning followed by lightly abrading constitutes an effective prebond treatment. Abrasion debris should be dry removed (without solvent), and care must be taken not too damage the thin primer layer.

3.2 Heating and Pressurisation

Heating and pressurisation are key installation issues since both can have a direct impact on the quality of the repair. Controlled heating is required to cure adhesives and cocure composite patches on the aircraft. Heating may be required with certain surface preparations and for adhesive primer cure. It may also be necessary for drying structure prior to repair installation. Pressure application is needed to mate the patch to the aircraft structure. Adequate pressure must be applied to ensure proper bondline thickness and minimise bondline voids and porosity. It also causes the adhesive to flow and properly "wet" the treated surfaces to achieve adequate adhesion. In the case of cocured composite patches, pressure may be required to consolidate the composite in order to obtain the desired mechanical properties.

Heating may be conducted by any of a number of methods provided they are able to safely control the temperature in the repair area within prescribed tolerances without contaminating the repair. Typical on-aircraft heating methods include electric-resistance heat blankets, infrared heat lamps and hot air devices. Application specifics determine the method best suited to a given repair. Heat blankets are typically used to cure adhesives, whereas heat lamps are usually the choice for silane drying and precuring primers. All heating devices must be controlled by some means so that heat can be applied when and to the levels required. This is particularly important for adhesive cure since controlled heat-up and cooling rates are usually prescribed. "Hot bonders" that automatically control heating based on temperature feedback from the repair area are normally used with heat blankets. These units or similar means can be employed to control heat lamps and hot air devices.

Attaining specified repair temperatures within desired tolerances can often be difficult on aircraft, since portions of the structure can act as "heat sinks." These regions conduct heat away from the repair site and become locally cooler, creating the possibility that the adhesive may not fully cure. Thermal surveys of the repair area are important to ensure proper heating will be attainable. The surveys should be conducted on the actual repair area using the equipment to be employed during the repair. They can determine the placement of insulation materials or the locations needed for supplemental heating, and they will reveal the required temperature readings for surrounding "monitoring" locations that can be used to determine the temperature in the repair area.

Pressure application on aircraft may also be achieved by a variety of means. These include vacuum bag, inflated bladder or various forms of mechanical pressure. The use of a vacuum bag is the most common since it is almost always the most convenient. Vacuum bags are light, conform to almost any surface, apply uniform pressure, can remove volatiles from the repair area and can hold a heat blanket in place. To apply pressure this way, a bag is built over the repair area and air is extracted, allowing atmospheric pressure to be applied. In most cases, it is not desirable to achieve a full vacuum throughout the cure cycle since the vacuum allows volatiles in the adhesive, such as moisture or solvents, to volatilise more readily, and it allows entrapped air to expand more easily. Often, high vacuum levels are applied initially to remove volatiles from the repair, then vacuum levels are reduced before the adhesive gels in order to minimise porosity in the bondline. Levels corresponding to 0.034-0.069 MPa pressure are common. Bladders inflated with air can be used to apply positive pressure (as opposed to vacuum) on a repair area. This may be desirable to minimise void formation due to the evolution of volatiles. However, this approach is not usually convenient since bladders must be held against the structure in some way and they require a frame or fixture to react against. If this fixture is fastened to the structure, pressure is limited to prevent damage. Mechanical pressure may also be applied by clamping or other means. Again, these forces must be reacted, and it may be difficult to apply uniform pressure over a large area.

Fibre composite patches should be precured under positive pressure per manufacturer's recommendations whenever possible. This is normally done in an autoclave to minimise porosity and achieve the per ply thickness value envisioned by the design. The cured patch can be nondestructively inspected prior to application on the aircraft. The knowledge that the patch has minimal porosity, typically a couple percent or less, eases the on-aircraft inspection burden for the adhesive bondline. At times, fibre composite patch materials are cocured on the aircraft with the adhesive. This may be done to allow them to conform to a complex geometry where the alternative of and secondarily bonding would require a tool to recreate the surface of the repair area. Prior to cocuring, fibre composite patches are often consolidated in an autoclave to minimise porosity and achieve the desired fibre volume.

3.3 Additional Installation Considerations

Although many successful repairs are accomplished on aircraft in the field environment, some care must be taken to control the repair site. Repairs should be conducted in aircraft hangers to provide protection from inclement weather. This environment allows for easier access to the repair area and to required equipment and facilities, such as power, air and vacuum source. Often, the repair must be shielded from air currents, especially when heating with infrared lamps. Temperature and humidity should also be controlled within reason. Temperatures should be in the 10°C to 32°C range with between 30% and 70% RH; a narrower range is desired. In all cases, the repair area must be protected from airborne and other contaminants. Cleanliness must be stressed for all adhesive-bonding operations. Prepared surfaces must not be touched, and handling of adhesives and patch materials should be minimised and conducted by personnel wearing appropriate gloves.

Adhesives, primers and surface preparation chemicals must be stored, handled and disposed of properly in order to achieve a successful repair and ensure worker safety. The material safety data sheets (MSDSs) for all materials must be read and understood. Local fire, safety and environmental regulations must be understood and followed, and appropriate personal protection equipment must be worn for some operations.

The typical repair film adhesives are stored frozen in sealed bags and are ready to use after warming to ambient temperature prior to bag opening to minimise the chances that moisture will condense on the material leading to porous bondlines during cure. Freezer storage allows for a shelf life typically between 6 months and one year. Time out of freezer storage must be monitored. This "out time" for repair adhesives depends on the temperature/humidity environment and is in the order of several days. Bond primers have similar reduced-temperature storage issues. Film adhesives generally contain a carrier cloth that helps control adhesive flow and final bondline thickness.

Many two-part paste adhesives have the advantage that they can be stored at ambient temperatures for extended periods and can cure, given enough time, at typical ambient temperatures. However, they have several limitations. In general, they do not possess the good overall mechanical properties obtained with epoxy-nitrile films. Also, the two parts must be mixed properly prior to use. Once mixed, the "pot life," generally less than 60 minutes, must not be exceeded prior to application. Bondline control is an issue that must be addressed. This and the mixing issue can be minimised by packaging schemes, including glass beads for bondline control, developed by the material supplier.

4. CERTIFICATION

Certification of any repair is an important process and must address a number of design, installation and in-service inspection issues. Unfortunately there is currently no widely accepted standard for the certification of bonded repairs. Repairs must be designed on the basis that the repair efficiency can be predicted and they should be designed conservatively with respect to the various failure modes to include the surrounding structure. In this regard, an approach similar to the one outlined by the USAF Structural Integrity Program identified in MIL-HDBK-1530 [19] can be used as the basis for certification.

For the purposes of this discussion, a distinction will be made between repairs to non-critical structure and those to safety-of-flight-critical structure. Additional certification requirements apply to the latter and these will be considered separately. For the former, the following list is indicative of the type of issues that should be examined in a certification program:

- The ability of the patch to operate under the environmental conditions that can be expected. This largely concerns issues such as the operating temperature and the choice of materials that were considered in the design. Some repairs that could be subjected to repeated impacts may need appropriate materials selection and possibly the design of a suitable protection system. Repairs that may be subjected to high-velocity airflow may also require appropriate sealing.
- The design of the repair with regards the expected operational loads. The design of the repair should show that the repair (patch and/or adhesive) will not fail under design ultimate load and the repair is not susceptible to any expected fatigue stresses. Note that a number of failure modes may be possible and these should be considered. If the repair is to be made to highly loaded structure, checks should be made on whether any growth of the defect may in turn cause damage (disbonding or delamination) within the patching system.
- The influence of the repair on the underlying structure. Checks need to be made to confirm for example, that the stress intensity at the defect, following repair, has been reduced to a level such that any crack growth rate is manageable. Repair installation must be performed using validated and appropriate materials and processes to ensure no additional damage is caused during the application.
- Setting of appropriate in-service inspection intervals. For noncritical repairs, even the complete failure of the repair will not compromise flight safety, however, inspections will be required to confirm the rate (if any) of subsequent growth of the underlying damage. NDI can also be used to confirm that the repair has been applied without any serious defects such as large bondline voids or disbonds.
- Quality control procedures have been met. These include the use of effective training methods for design and installation staff, the use of qualified materials that are in-life, the use of validated design and application procedures and the control of the repair environment to ensure full cure of the adhesive.

Where repairs are considered for safety-of-flight-critical structure, some additional considerations are likely to apply, as mentioned above. Chief amongst these is the issue of long-term bond durability in the service environment, particularly moisture, and the current inability to satisfactorily predict or measure this durability. NDI methods are able to measure the presence of defects within an adhesive bondline, however this is necessary but not sufficient for assuring bondline integrity. No NDI method is currently able to measure the strength of an adhesive bond and therefore (through repeated measurements over time) any gradual degradation in the strength. For this reason, a fail-safe approach is often currently adopted for flight critical structure. This does not allow full credit to be given to the repair for restoring residual strength and reducing the fatigue crack growth rate [20]. The assumption is therefore made that the repair could fail at any time by adhesive bond failure and certification of the repair is performed on this basis. Clearly this is very conservative as the requirement is then that the damaged, unrepaired structure has to be capable of withstanding design limit load in the absence of the repair. While this is a safe approach that minimizes certification requirements, it also prevents many cost effective bonded repairs from being considered.

At this point it is important to clearly state that properly designed and installed adhesively bonded repairs have a most impressive track record for outstanding long-term durability [2,3]. Unfortunately, even a well-designed repair can suffer from adhesive bond durability if is not applied in accordance with approved procedures. For this reason strict adherence with a comprehensive and validated quality plan is necessary to ensure the long-term durability of the bond.

There are at least three approaches that could be used in the future to overcome this problem, and one of these involves the use of a simple accelerated test method to predict the long-term behaviour of the repair. A simple adhesive "witness coupon" can be made alongside the actual repair using the same materials and processes. If it can be shown that an accelerated test on such a coupon accurately reflects the actual long-term behaviour, this would provide confidence in the durability. The other two approaches will require further research to be effective and are considered in the final section of this paper.

5. REPAIR EXAMPLES

5.1 F-16 Lower Wing Skin "Vent Hole" Repair [21]

Fatigue cracks, running forward and aft, initiated at fasteners around the vent holes in lower left wing skins in older F-16 aircraft, Figure 3. These holes vent fuel from the fuselage via a tube that is attached to the wing skin by two concentric rows of fasteners through a flange on the tube. Some cracks extended to the second fastener row. Cracks extending beyond the vent tube flange would allow a direct path to the wing fuel tank and result in a fuel leak. This was considered the end of the service life for the lower wing skin.

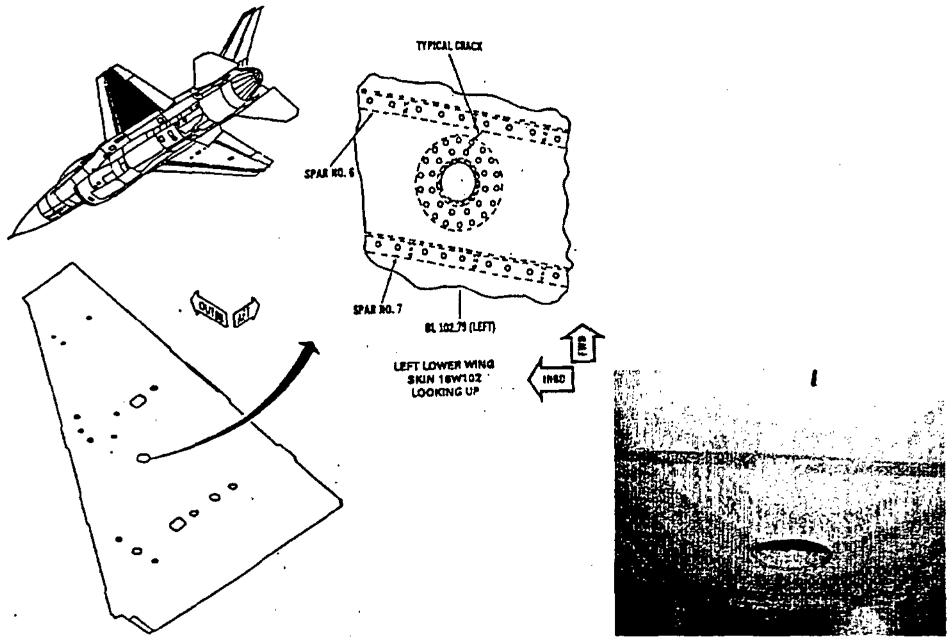

Figure 3: *Typical Crack Location in F-16 and, inset a photograph of an installed repair patch.*

A mechanically fastened aluminium patch repair was designed for the application. Analysis showed this repair would not stop fatigue crack growth but would extend service life from about 3300 hours to about 5700 hours, but short of the 8000-hour goal. Installation of the mechanically fastened repair would have been time consuming since the upper wing skin must be removed to permit access to the required fasteners in the lower wing skin. The need to drill new fastener holes in the skin was also a very significant disadvantage of this approach.

A bonded fibre composite patch repair was considered the best option since it could be performed without drilling additional holes or removing the upper wing skin. F-16 engineering at the USAF Ogden Air Logistics Center at Hill AFB teamed with engineers at AFRL to design and install bonded boron/epoxy patches to repair the vent hole cracks. F-16 engineering designed the repair and AFRL installed the patches.

The starting point for the composite patch design was to match the stiffness of the initial aluminium doubler design. Finite element analyses were conducted using the IDEAS code for both the wing and the patch. Before the patch design was completely optimised, an immediate opportunity arose to test the repair on two vent hole subcomponents. The results of the fatigue tests indicated the repair would meet the life extension goals for the repair, and further patch optimisation was discontinued. The precured patch consists of 14 unidirectional plies of boron/epoxy, to be aligned on the structure normal to the fatigue cracks, and ±45° plies on the top and bottom of the patch. After cure and prior to installation, a hole is established in the patch for the vent. The patch covers existing fasteners in the structure.

The important metal surface preparation step was initially grit-blast/silane (GBS) with application of Cytec Fiberite BR 127 primer which was cured prior to the application of the adhesive. The latest repairs were installed using grit-blast/sol-gel with Cytec Fiberite BR 6747-1 primer cocured with the adhesive, reducing repair time by over 3 hours compared to the GBS process. In order to keep cure temperatures reasonable for on-aircraft application while meeting the service temperature requirements, FM 87-1 and FM 300-2 adhesives from Cytec Fiberite were selected as repair materials with on-aircraft cures in the range of 104°C to 127°C. Extensive thermal surveys on an F-16 wing were used to determine the heating methods and necessary insulation. Electric-resistance heat blankets cure the adhesive while infrared heat lamps are used for silane drying and primer cure. Pressure is applied via vacuum bag. The fuel tanks are air purged during the repair operations and the lower explosion limit (LEL) is maintained well below that required for safety.

The first F-16 vent hole bonded repair installation was completed in early 1993. Only this initial repair utilized FM 87-1. That repair patch material was Textron 5505 boron/epoxy cured at 177°C. Most subsequent repairs were conducted using Textron 5521 cured at 121°C with FM 300-2 adhesive. Twenty aircraft from three countries have been successfully repaired. To date, there are no known problems with the repair installations, and nondestructive inspections of cracks beneath the patches reveal no concerns.

5.2 RAAF F-111 Lower Wing Skin Repair

F-111 aircraft in service with the Royal Australian Air Force (RAAF) have recently been found to suffer from fatigue cracking in the outboard section of the aluminium lower wing skin [22]. The cracking is caused by a stress concentration from a runout in the forward auxiliary spar to create a fuel flow passage. When the first crack was discovered, fracture mechanics calculations indicated that it was beyond critical length at design limit load. A conventional mechanically fastened metallic repair was considered, but this was unattractive from an aerodynamic standpoint (excessive thickness). New fastener holes would not have been acceptable in this highly stressed primary structure, and the crack would have been uninspectable beneath such a repair. A bonded composite repair was the only alternative to scrapping the wing. It must be emphasised that because of its criticality this repair is not a typical example, but rather represents the limit of what bonded repair technology can achieve. Because of certification concerns [20], repairs to critical defects in primary aircraft structure are unlikely to become commonplace in the near future, and in this case an extensive program was required to certify this repair.

Extensive and detailed 2-D and 3-D finite element analysis was conducted so that the stress distribution around the defect could be quantified. This revealed that the wing skin at this location was subject to secondary bending and this was the explanation for the observation that the crack had initiated on the inside surface of the wing skin. The model was validated with strain measurements from a full-scale wing test undertaken at the Aeronautical and Maritime Research Laboratory. In addition, three levels of specimen testing were undertaken:

- Small, inexpensive coupon-sized specimens were used to investigate the effects of impact damage, temperature and moisture and load spectrum truncation effects.
- Panel specimens with a full-scale representation of the local wing geometry were used as structural details in a fatigue and environmental study.
- Large box specimens were used to represent the wing as a quasi full-scale test article in testing static and fatigue strength and an examination of thermal residual stresses.

A repair was designed using boron/epoxy as the repair material as this provides lower levels of thermally induced residual stress compared with graphite/epoxy and enables the ready use of eddy current NDI methods to confirm the length of the crack that was left in the wing. Cytec Fiberite FM 73 epoxy adhesive was selected and cured at the comparatively low temperature of 80°C to minimise the thermally induced residual stresses [23]. This cure cycle had previously been carefully validated for another complex repair [24]. The surface treatment used was the GBS process described above. Advantage was taken of nearby hard-points on the wing to make use of positive pressure during the cure. An inflated bladder was used to apply pressure to the repair and the pressurisation loads were reacted out via a rigid plate to the hard-points. A similar system was used in the earlier application of doublers to the upper surface of F-111 wing pivot fittings [24].

This lower wing skin repair was predicted to, and subsequently proven in service for around 3 years to, reduce substantially the crack growth rate of the defect. This wing was later used as a fatigue test article and survived for over 5000 hours of F-111 spectrum loading with no further detectable crack growth. It was also noted that a test wing with a much smaller crack in this region failed from the crack in less than 1000 hours of simulated flights.

As most wings in the RAAF fleet have not yet developed cracks, these repairs are currently being applied to the fleet as preventative reinforcements to prevent the initiation of cracks in the future. This is an excellent example of how the technology can be used to extend the life of airframes. The repair to the wing is shown in Figure 4.

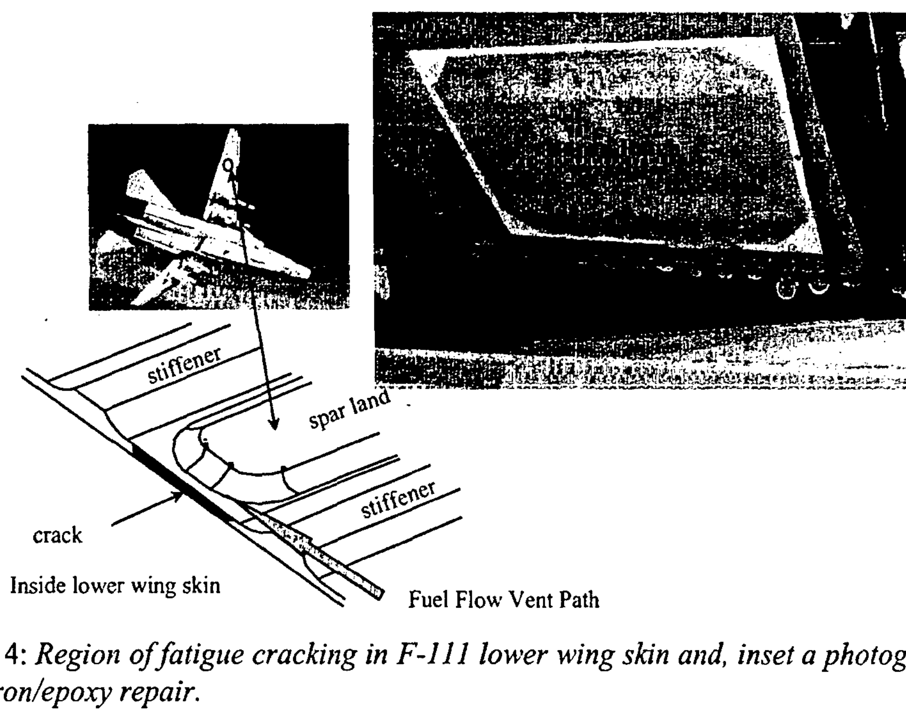

Figure 4: *Region of fatigue cracking in F-111 lower wing skin and, inset a photograph of the boron/epoxy repair.*

6. FUTURE TECHNOLOGY REQUIREMENTS

While this technology is now reasonably mature and is routinely used for aircraft repairs, further research is required to address certification concerns for complex repairs, to further simplify design and application procedures and to extend the capability to address damaged structure that cannot currently be repaired. Recently, three major reviews were undertaken independently to define the general R&D needs of bonded repair technology. These reviews were by the Committee on Aging of U.S. Air Force Aircraft in 1997, The Technical Cooperation Program (TTCP) Aeronautical Vehicles Action Group on Certification of Bonded Structure [25] in 1999 and an Australian Defence Science and Technology Organisation (DSTO) strategic review in 1998. This section aims to summarise some of the key findings from these reviews.

R&D proposals are grouped here into Design, Certification and Application areas. Although there is some overlap, these are useful groupings for this discussion. In the design area, there is scope to simplify routine design procedures through the use of validated software design packages for personal computers; the USAF has work in progress on a package called CalcuRep, as mentioned earlier. Expert systems could also be of use in the assessment of damaged structure and design analysis. There is scope to optimise the stress distributions within repairs using routines within Finite Element Packages. Work in Australia [26] has shown that it is theoretically possible to reduce some of the peak stresses in bonded joints through careful shaping of the adherends.

The scope of the technology can be increased if design procedures can be developed for repairs to more complex structure such as thick, highly stressed components or highly curved components. There is a limit to the amount of load that can be reliably transferred through an adhesive joint. For very thick structure with severe damage, bolted repairs may be the more suitable alternative. There are however, some limitations that may be overcome by further research. In a curved repair, stresses are developed normal to the plane of the repair (peel stresses) and current understanding of these and the associated design allowables for various repair materials is poor. Repair designs for corrosion and, more particularly, for acoustically generated fatigue stresses are other areas where there is scope for improvement. The nature of acoustic fatigue and the need for dampening of the structure means that an entirely different repair philosophy is usually required. Both Australia and the U.S. have active programs in this area [27,28]. Finally, the ability to calculate more readily the magnitude of any thermally generated residual stresses would be advantageous for some repairs. The problem is that it is difficult to estimate accurately the degree of thermal expansion of the repaired area due to the constraint that is provided by the surrounding cooler structure.

The certification issues requiring further research are dominated by the need to provide confidence in the long-term durability of the adhesive bond. While it is well known that well-designed and produced adhesive joints do have excellent durability, this is only the case if they are prepared using well-documented and validated procedures. There is a need for a simple accelerated test method that can be used to indicate the likely long-term performance of the joint. Such a method could either be used as a witness specimen during a repair (prepared at the same time and with the same materials/processes) or as a laboratory method to indicate the likely effect of changes to procedures. Any such test method will, of course, need to be validated against actual long-term performance and in this regard the wedge test is particularly attractive [20].

Models currently exist which can be used to predict the rate of continued damage growth after repair, and this will be required in some cases for certification of repairs to primary structure. Validation of these models is required and further development of them to ensure that they can accurately predict the effect of various parameters such as the nature of the loading (spectrum and R ratio effects), the influence of residual stresses and environmental effects.

Certification issues also arise when repairs are required to materials other than the standard aluminium alloys, for example titanium, nickel or stainless steel. Considerable work is required to ensure that acceptable levels of bond durability can be achieved and the availability of the validated accelerated test method mentioned above would be of considerable help. Related issues include bonding over fasteners where it may be necessary to prove that the repair will not be compromised by fastener movement or durability problems caused by the dissimilar materials and surface preparation methods. The development of either a Smart Patch or a post-bond NDI method would help certification concerns [20]. Unfortunately, the scientific problems with the NDI method are considerable, and it is unlikely that a one will be available in the near future that can measure the level of bond strength as described in the previous section. Of more immediate hope is the development of a Smart Patch that is able to sense its state of health by virtue of embedded microelectronic sensors. Such a patch has been developed and is currently being flight tested on a RAAF F/A-18 [29]. It is intended that these patches will be self-powered using piezoelectric elements to "harvest" power from the parent structure. Future developments will also include active patches. Using piezoelectric sensors and actuators, these patches will, for example, be able to provide active damping as well as reinforcement to counter acoustic fatigue.

Finally, in the area of repair application, research is required to develop new methods of preparing metallic surfaces for adhesive bonding. While current methods are extremely effective, improvements would lead to further reductions in repair time (and hence cost) as well enabling repairs to be applied with reduced levels of quality assurance. This would assist for field-level repairs or perhaps battle damage type repairs. In the area of materials, rapid screening methods would be helpful to quickly assess the likely potential of the material for use in repairs. Currently, large and expensive test programs are required to generate B-basis allowables for new materials. Methods that provide the required design data at reasonable cost are required. In the area of NDI, there is scope for the development in the short term of methods that can indicate if surface preparation methods have been correctly applied. While not in itself a guarantee of long-term bond durability, such NDI methods would be helpful in a risk management context.

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